

Sailplane Wing Box Design by Use of Graphite/Aramide/Epoxy Material

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Introduction

In the 1970's particular attention was paid to advanced composite materials for future aerospace structures because of the potential mass savings they can offer. This objective could be very important in the soaring flight too for three main reasons: for high performance sailplanes a larger wing loading variation should be available between the minimum and maximum values, improving the flight performance; for powered sailplanes engine power could be reduced with consequent beneficial effect for fuel savings; for training sailplanes a reduced minimum wing loading should permit flight with lower sinking speed.

In comparison with glass fibre reinforced plastic, advanced composite materials (graphite/epoxy and lower aramide/epoxy) present higher Young's and shear moduli leading to higher bending and torsional stiffness of the wing.

Mass savings in the range of about 11 to 24% have been obtained on sailplane wings by using advanced composite materials instead of some GFRP elements [1]. In ref. 2 the theoretical mass savings of graphite/epoxy stiffened panels is up to 50% of the aluminium stiffened panels on the basis of data presented by NACA, Istituto di Progetto di Aeromobili, Politecnico di Torino, Italy. In practice savings of 32 to 42% have been obtained.

A computer program has been designed to evaluate the mass optimization of hat stiffened panels [3]; however, owing to the low aerofoil thickness of sail plane wings, the utilization of such panels (and also open-section stiffened panels) would not be expected to give the same good structural efficiency. For this application, the honeycomb sandwich panels give better efficiency.

A purpose of this paper is to show the design and fabrication process of a new advanced composite cellular wing structure. Being a part of a wider pro-

gram leading to the realization of a "Very Advanced Technology Light Twin" aeroplane, it has been developed during research carried on at the "Delft University of Technology, Department of Aerospace Engineering", in cooperation with the technical staff and colleagues of the "Composite Materials Work Group" [4].

A wing box 1300 mm long, 600 mm wide and 160 mm high has been manu-

factured in graphite/aramide/epoxy material and has been planned to be tested under shear/bending loads (fig. 1).

Previously, a flat cellular panel 500 mm long and 155 mm wide was manufactured in graphite/epoxy by thermal expansion moulding process for the purpose of getting acquainted with fabrication techniques (fig. 2).

The question arises: if such a structure should be of interest in the construction of a sailplane wing. In the absence of detailed data about current CFRP sailplane wings on mass, production costs, maintenance and repair costs, etc., the author is not able to give a positive answer. Nevertheless, in case it is of some interest, the design of a sailplane wing is reported; it is related to the modified 15 meter M-300 glider conceived for an extruded wing [5] with a new minimum wing loading of 23 kg/m² and unchanged maximum value of 39 kg/m².

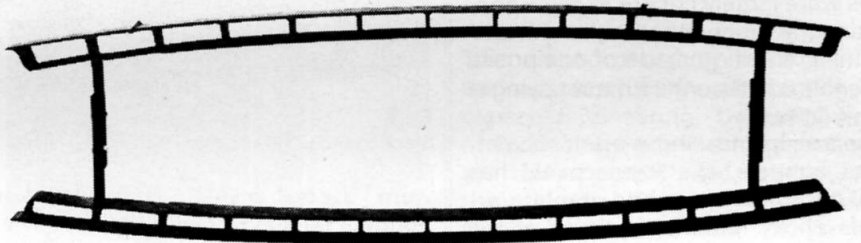
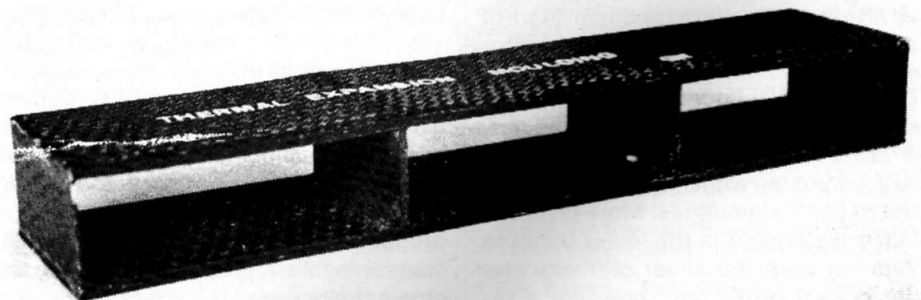
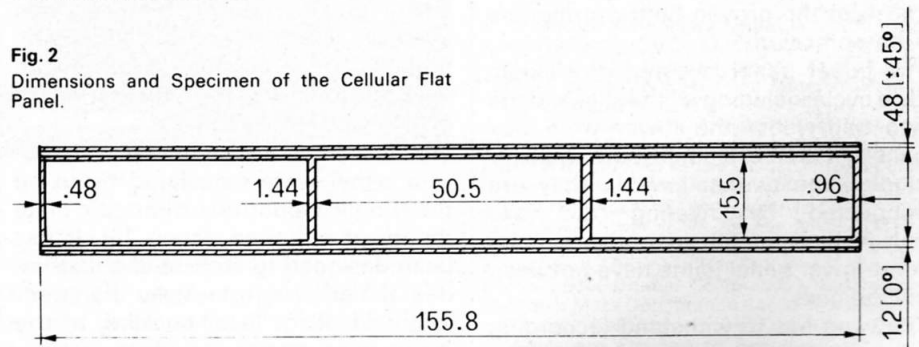


Fig. 1
Cellular Wing Box Specimen.

Fig. 2
Dimensions and Specimen of the Cellular Flat Panel.



Preliminary design

In the design of composite structures, the main advantage should come from realizing the structure with very few components, each one of them manufactured in a single cure cycle.

A possible typical cross-section of the wing is sketched in figure 3; the central box will carry all bending loads and most of the shear/torsion loads, supplemented in the case of the latter by the leading edge box. The central box has been designed like a cellular structure leading to the realization of an integral ribless wing.

An extruded aluminium alloy wing box has been manufactured first with such shape [5] and a shear bending test was performed on a 7.67 m long wing which only had two ribs located at the wing/fuselage connection. The experiment demonstrated the capability of this structure to withstand nearly the full prescribed loads, with special regard to the crushing loads on the webs; the failure occurred at a very high load factor ($n = 8.72$) in the central part of the structure where the webs were largely cut out to allow connecting the fittings.

A similar situation made of composite materials could permit a mass saving of about 50%.

Since a sailplane wing is often subjected to impacts, the lower panel has been designed as a hybrid graphite/aramide/epoxy material because of the good impact resistance of aramide fibres; the upper panel has been designed in graphite/epoxy material because of the proven better properties in compression.

The upper panel is cured in a single cure cycle including all the skin and the two half-webs; the lower wing box panel is cured in a single cure cycle including the two half-webs; they are connected by riveting the two half-webs along the span; the rear and front-lower panel joints have not been yet defined.

The wing has to withstand (according to the OSTIV Airworthiness Requirements) an ultimate bending moment of 35 kNm and an ultimate shear load of 9,5 kN, corresponding to a load factor $n = 10$.

The stacking sequence of the panels is sketched in figure 4. The inner and outer skins were designed as a $(\pm 45^\circ, 0^\circ, 90^\circ)$ laminate, while the bridges consist of $(\pm 45^\circ)$ laminates. More layers of fabric were used in the lower panel to improve both the shear stiffness and the impact protection, however a reduced number of unidirectional layers

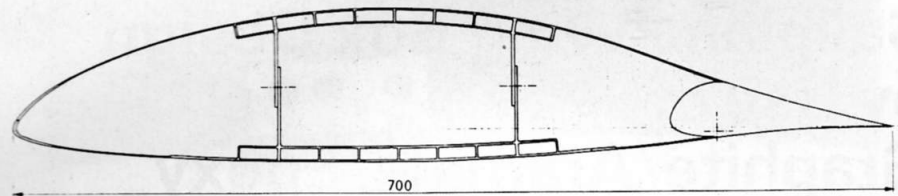


Fig. 3
Advanced Composite Sailplane Wing Cross-Section.

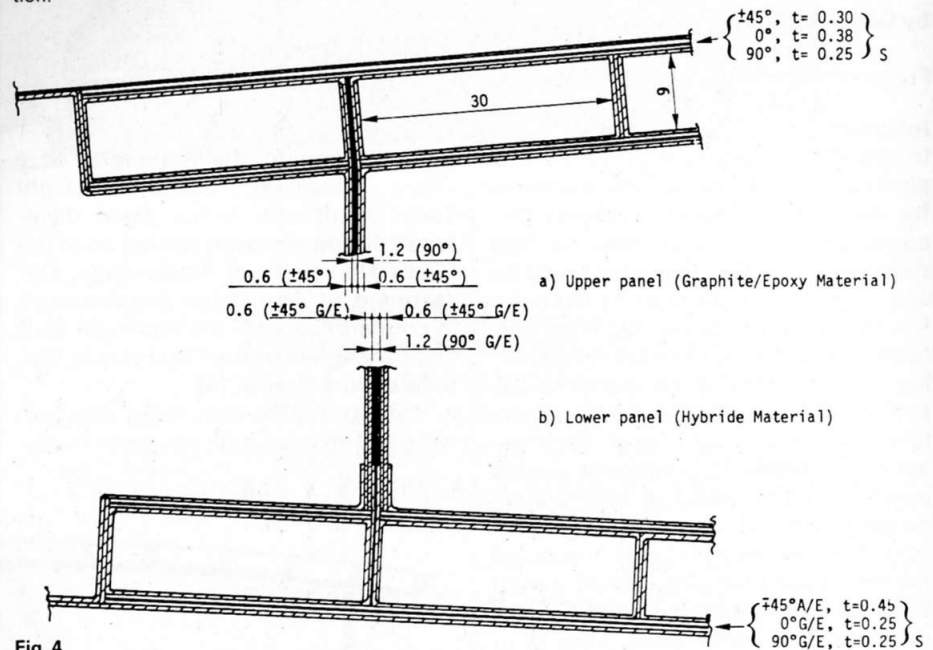


Fig. 4
Stacking sequence and dimensions of the panels.

were used because of the higher Young's tensile modulus in comparison to the compressive modulus; in this way the wing box was designed with a structural symmetry. A $(\pm 45^\circ, 90^\circ, \pm 45^\circ)$ graphite/epoxy material was used in all half-webs for improving both the critical crushing loads and the bearing strength in the rivet holes (50% of $\pm 45^\circ$ and 50% of 90°).

The buckling stresses of the compression panel were calculated by using the simply supported orthotropic plate theory of stiffened panels [6]. It has been designed to withstand a load index of 1350 N/mm to make the structure critical for local buckling of the laminate between two bridges.

The theoretical mass of a 9 meter untapered central wing box is 27 kg.

Material selection

Since the temperature of 54°C is normally not exceeded, the resins system used for sailplane construction is cured at room temperature and post-cured at 60°C ; the fabric is usually impregnated with resin by hand and several layers are applied in order to obtain the required thickness.

A pre-preg layer is, instead, used for

aeroplane structures of advanced composite materials and cured, at least, at 120°C . Of course this process is more expensive than the first one; however, some advantages should result:

- better mechanical properties at high temperatures
- better alignment of the fibres and closer control of the resin content
- reduced time during the lay-up process
- less adverse effects on the workers health.

The material used in manufacturing the above-mentioned wing box was the unidirectional graphite/epoxy T300/F550 and the square fabrics graphite/epoxy T6341/R2503 and aramide/epoxy kevlar 49/R2503; they were pre-pregged aiming at a cured fibre volume content of 60%.

The elastic mechanical properties of the unidirectional material are shown in figure 5. The discontinuity is due to the use of different specimens for the tensile and compression test; however, when testing compression specimens starting from a slight tensile load no discontinuity at zero load was observed [7].

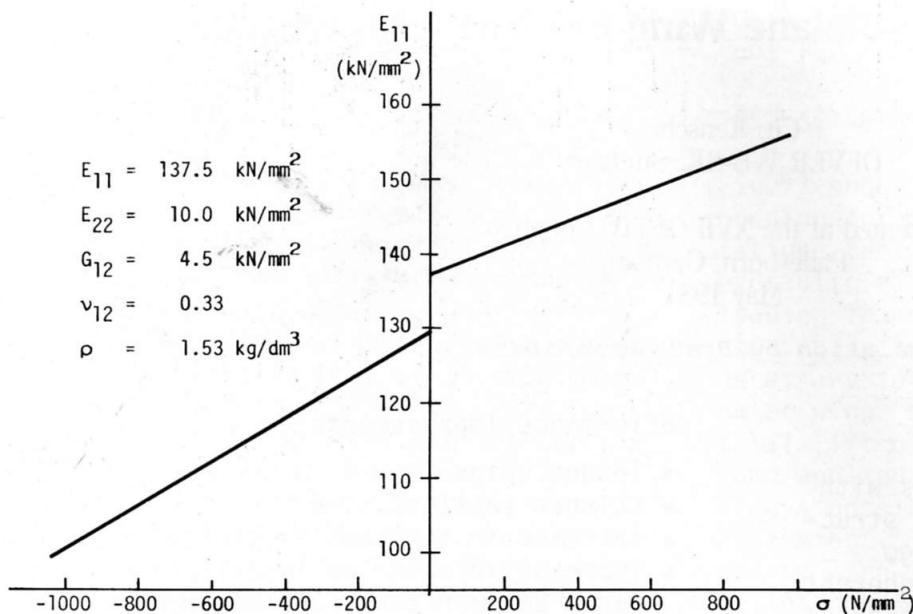


Fig. 5
Young's Modulus/Stress Curve for Unidirectional Material T300/F550 Hexcel.

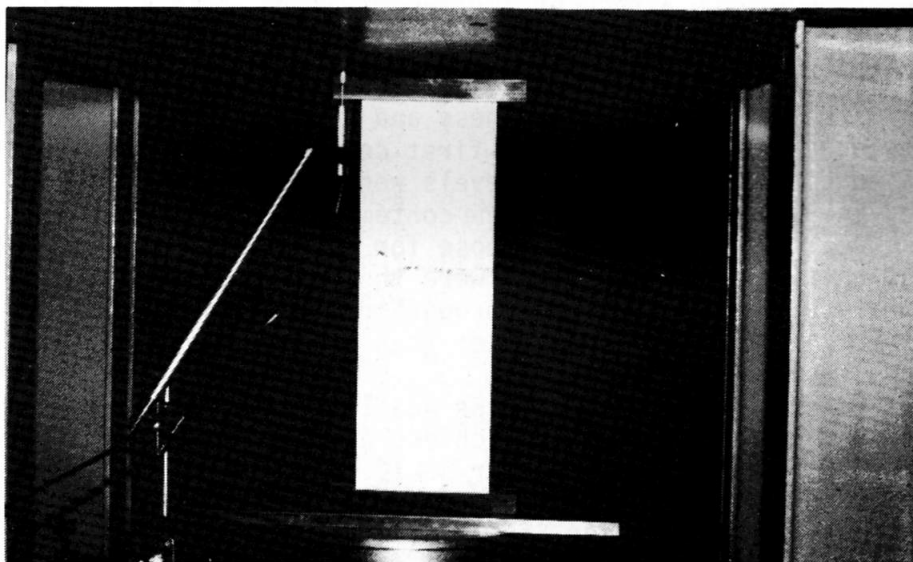


Fig. 6
Compression failure of the Cellular Flat Panel.

Production process

In an attempt to avoid the use of an autoclave process, some experiments have been carried out with a thermal expansion moulding technique and the panel of figure 2 has been produced. Indeed, it became clear that, by this process, the pressure raise caused by solid expanding rubbers is very badly controlled by the temperature because there will always be an unpredictable volume gap in the closed mould; consequently this technique has been rejected.

The compression test carried on this panel to check the capability to withstand the design compression stress

without local buckling failure has not been fully satisfactory; the failure occurred at a load of 150 kN by local buckling of the laminate between two bridges, followed by an explosive delamination of the skin (fig. 6). However, telegraphing effects and initial imperfections were present in the laminate and both probably have caused a reduction of 25% of the local buckling load.

Better results should be possible by using the inflated rubber technique [8, 2]. However, for its simplicity the panels of the wing box of figure 1 have been manufactured on a mould that, combined with solid silicon rubber cores,

was suitable for an autoclave controlled pressure cycle using a vacuum bag (fig. 7).

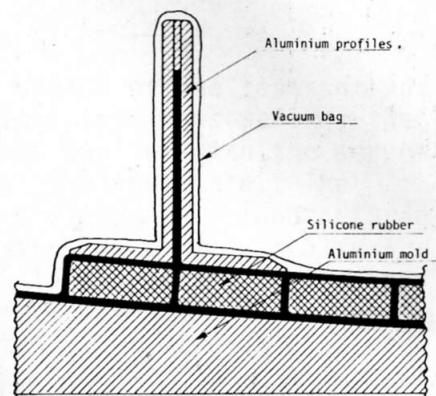


Fig. 7
Vacuum bag stacking sequence.

In the sailplane production the main disadvantage of this process would be the high cost of a long autoclave; therefore, from this point of view, the inflated rubber technique has been preferred since the rubber expansion is controlled more easily.

Acknowledgements

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